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TURBINE AIRFOIL FILM COOLING*

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INTRODUCTION

Emphasis is continuing to be placed on developing more accurate analytical models for predicting turbine airfoil external heat transfer rates. Performance goals of new engines require highly refined, accurate design tools to meet durability requirements. In order to obtain improvements in analytical capabilities, programs are required which focus on enhancing analytical techniques through verification of new models by comparison with relevant experimental data. The objectives of the current program are to develop an analytical approach, based on boundary layer theory, for predicting the effects of airfoil film cooling on downstream heat transfer rates and to verify the resulting analytical method by comparison of predictions with hot cascade data obtained under this program.

BACKGROUND

The overall approach to attaining the stated objective has involved a series of three programs. The initial program, performed under Contract NAS3-22761, assessed the capability of available modeling techniques to predict non-film cooled airfoil surface heat transfer distributions, acquired experimental data as needed for model verification, and provided verified improvements in the analytical models. This effort resulted in a baseline predictive capability and was reported in CR 168015 (ref. 1) published in May 1983.

The problem of heat transfer predictions with film cooling was broken into sequential efforts with the effect of leading edge showerhead film cooling being investigated first, followed by a program to study the effects of the addition of discrete site suction and pressure surface injection. The effort on showerhead film cooling was performed under Contract NAS3-23695 and was reported in CR 174827 (ref.2) published in July 1985. As part of that program, a five-row, simulated common plenum showerhead geometry was tested to determine differences between film and non-film cooled heat transfer coefficient distributions downstream of a leading edge, multiple hole film cooling array. Building on non-film cooling modeling improvements incorporated in a modified version of the STAN5 boundary layer code developed under Contract NAS3-22761, a program was developed to analytically model and predict differences resulting from leading edge mass injection.

The current program, being performed under Contract NAS3-24619, is intended to extend the analytical code development to include discrete site pressure and suction surface injection, with and without leading edge blowing, and to obtain relevant hot cascade data to verify the model improvements.

*This work is being performed under Contract NAS 3-24619

PROGRESS

Because of the long lead times associated with hardware design and fabrication, initial efforts on the program have been aimed at the experimental phase. The analytical efforts under the program are just beginning with the design mode analysis phase having started in August. This phase is intended to demonstrate the use of the base boundary layer method in a film-cooled turbine airfoil design system environment. This initial study is addressing details involved with method set-up procedures (e.g. defining initial and boundary conditions) and the qualitative behavior of the film cooling models for a relevant film-cooled airfoil design.

The experimental phase of the program will be an extension of the previous contract work. The hot cascade tests will utilize the same facility, cascade and experimental techniques used in the previous contract, with the instrumented airfoil in the cascade replaced with one containing suction surface and pressure surface film cooling arrays in addition to a leading edge showerhead film cooling array. Design of the film cooling arrays for the new airfoil has been completed and the fabrication and instrumentation of the airfoil is nearly complete.

The airfoil cooling design incorporates three separate film cooling supply plenums. One plenum will supply an array of leading edge showerhead film cooling holes. The geometry of this film cooling hole array will be identical to that utilized in Contract NAS3-23695. Two additional coolant supply plenums will be incorporated into the vane; one to supply an array of holes on the suction surface and the other to supply an array of holes on the pressure surface of the airfoil. The three separate plenums will allow independent control of the flow to each region of the airfoil.

The suction surface array will contain two staggered rows of holes centered at approximately 25.2% of the suction surface length from the leading edge. Based on the heat transfer results of the previous contract (NAS3-23695), this location will place the array midway between the points of boundary layer transition origin for the highest and lowest Reynolds number (Re) cases studied. This will result in the ability to move the boundary layer transition origin point across the array as the cascade operating conditions are changed. It is likely that the film cooling holes will act as trips, resulting in earlier transition for the low Reynolds number case.

In Figure 1, the two vertical lines between twenty and forty percent surface distance on the suction surface represent the location of the two rows of film cooling holes. This figure illustrates the position of the holes relative to the heat transfer coefficient distribution and clearly shows the suction surface holes located midway between the boundary layer transition origin points for the highest and lowest Reynolds number cases.

The location of the two rows of holes on the pressure surface are indicated by the two vertical lines on the pressure side in Figure 1. This array will be centered at 22.5% of the pressure surface length from the leading edge. As can be seen from Figure 1, this position is centered in the area of minimum heat transfer, at the point where the heat transfer begins to increase as you progress toward the trailing edge.

Details of the geometry for the film cooling arrays are summarized in Table 1. All hole diameters will be the same and will be identical to the showerhead hole diameters in the previous contract. The geometry of the showerhead array will be identical to that tested in the previous contract. Both the suction and pressure surface arrays will consist of two staggered rows with row spacing-to-diameter ratios of 4.0. The hole spacing-to-diameter ratio of 3.0 will be the same on both the pressure and suction surfaces. The injection angle in the leading edge will be the same as in the previous contract where the holes were aligned normal to the surface in the chordwise direction and at a 45^{o} angle in the spanwise direction. The suction surface holes will be inclined at 35^{o} to the surface in the chordwise direction while the pressure surface holes will be at 20^{o} in the chordwise direction. Both arrays will be normal to the surface in the spanwise direction. These injection angles and the location of the film cooling arrays on the airfoil are illustrated in Figure 2.

Also shown in Figure 2 is the thermal barrier cutout region. This cut thermally isolates the film cooling supply plenums from the regions of the airfoil where heat transfer measurements are made, similar to the technique in the previous contract. The retaining bar shown in Figure 2 ensures that the airfoil profile is properly maintained after the thermal barrier cut is made.

PROGRAM PLANS

Fabrication and instrumentation of the film cooled airfoil should be completed late this calendar year. Cascade testing is currently planned to begin in late November and will extend through June 1987. Work on the analytical effort will include prediction of cascade results during the design mode analytical phase which is currently underway. While the cascade testing program is being performed, the analytical effort will focus on a methods characterization study which will determine the qualitative/quantitative capabilities of the proposed analytical method by comparing analytical predictions with experimental results from the cascade tests. Following completion of this effort, the analytical task will enter the method refinement/verification phase. This will address modeling deficiencies revealed in the first two phases of the analytical program and will develop an improved analytical code that will be verified by comparison with the experimental data obtained during the course of the contract.

REFERENCES

1. L. D. Hylton, M. S. Mihelc, E. R. Turner, D. A. Nealy, and R. E. York, "Analytical and Experimental Evaluation of the Heat Transfer Distribution over the Surfaces of Turbine Vanes", NASA CR-168015, May 1983.

2. E. R. Turner, M. D. Wilson, L. D. Hylton, and R. M. Kaufman, "Turbine Vane External Heat Transfer", Vol I, NASA CR-174827, July 1985.

TABLE 1. GEOMETRY OF FILM COOLING ARRAYS

	LEADING EDGE	SUCTION SURFACE	PRESSURE SURFACE
HOLE DIAMETER (IN.)	0.039	0.039	0.039
(CM.)	0.100	0.100	0.100
HOLE SPACING/DIAMETER	7.5	3.0	3.0
ROW SPACING/DIAMETER	4.0	4.0	4.0
HOLE ANGLE	45^o (Spanwise)	35^o (Chordwise)	20^o (Chordwise)
STAGGERED ROWS	Yes	Yes	Yes
LOCATION OF CENTER OF ARRAY (% SURFACE DISTANCE)		25.2	22.5

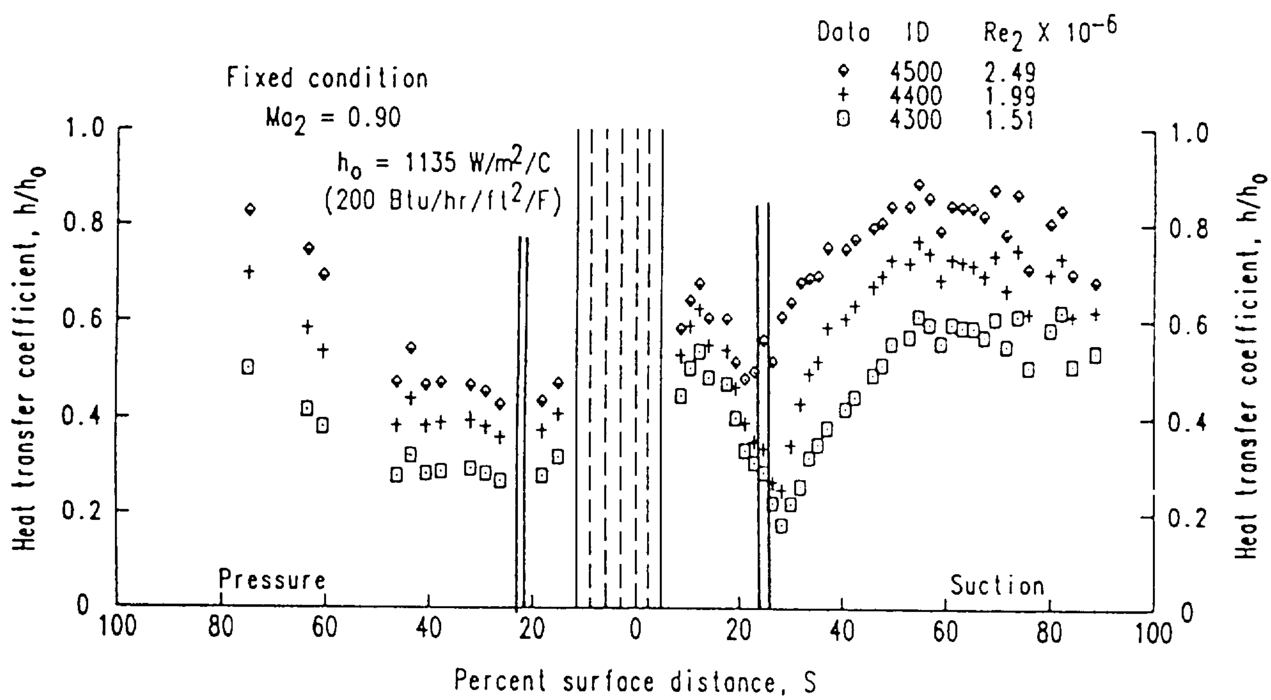

Figure 1. The effect of the exit Reynolds number variation on the C3X vane heat transfer coefficient distribution.

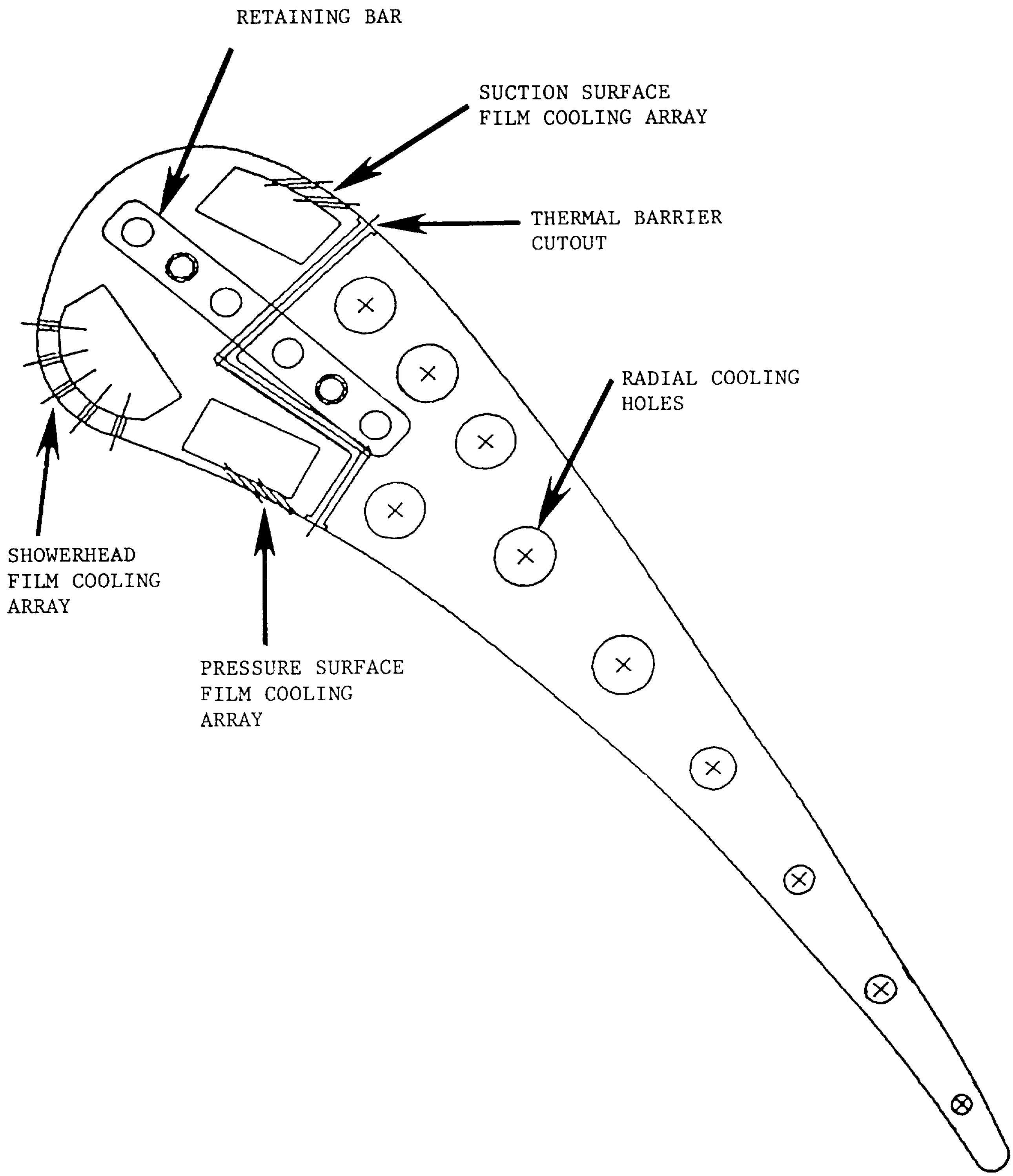

Figure 2. Airfoil geometry